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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MSC INTERNAL NOTE NO. 68-FM-82

March 29, 1968

03/29/1968

COMPARISON OF VENUS SWINGBY
AND MINIMUM-ENERGY TECHNIQUES
FOR MANNED MARS ORBITING

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MISSIONS

By James J. Taylor,

Advanced Mission Design Branch

MISSION PLANNING AND ANALYSIS DIVISION



MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

(NASA-TM-X-69721) COMPARISON OF VENUS
SWINGBY AND MINIMUM-ENERGY TECHNIQUES FOR
MANNED MARS ORBITING MISSIONS (NASA)

41 p

N74-70650

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COMPARISON OF VENUS SWINGBY AND MINIMUM-ENERGY
TECHNIQUES FOR MANNED MARS ORBITING MISSIONS

By James J. Taylor

SUMMARY

A study of manned Mars-orbiting missions that swing by Venus on either the inbound or outbound leg has been made for departures in the 1977 through 1985 cycle. Velocity requirements and scientific payloads were compared with those of minimum-energy manned Mars orbiting missions.

The propulsive velocity requirements for Mars orbiting missions which use the Venus swingby technique are shown to be greater than those for the minimum-energy missions by approximately 6000 fps. The total mission duration is approximately 10 months less than the minimum-energy missions. However, this reduction in mission time is achieved by reducing the stay time in Mars orbit by an equivalent amount (i.e., 10 months). The Earth entry velocity, although low for some opportunities is 10 000 fps greater than the minimum-energy missions for those swingby opportunities which present modest spacecraft propulsion requirements.

The scientific payload deliverable to the Martian orbit is compared with that of the minimum-energy missions by assuming equivalent spacecraft weights. An assembly of four uprated Saturn V's capable of orbiting 400 000 lb about the Earth at 262-n. mi. altitude is sufficient to place 11 000 lb of scientific payload in Martian orbit and to inject 1000 lb of it to Earth. The 11 000-lb orbital payload is about 10 percent of the payload capability of the minimum-energy mission.

INTRODUCTION

Manned Mars orbiting missions that swing by Venus have received considerable study (refs. 1, 2, 3, and 4) because of the potentially advantageous trade-off between mission duration and energy requirements.

The purpose of the analysis presented is to use the information in reference 3 to determine the propulsive velocity and spacecraft weights required for the Venus swingby missions and to compare the results with similar results of the minimum-energy manned Mars orbiting missions

described in reference 6. Reference 3 lists parameters of the Venus swing-by trajectories as functions of total mission duration, stay time in Mars orbit, Earth departure date, and Mars arrival date for inbound and outbound Venus swingbys.

The minimum-energy manned Mars orbiting mission consists of two near-Hohmann planetary transfers with an appropriate wait time between transfers. The "appropriate" wait time is approximately 300 days, and the total mission time is about 950 days. Any deviation to this flight plan, such as a Venus swingby, results in increased energy requirements. However, the reduced mission time associated with the Venus swingby may be desirable.

ANALYSIS AND RESULTS

Mission Profile

Figure 1 is a schematic of the heliocentric phase of an outbound swingby and an inbound swingby. The Venus encounter bends the trajectory so as to reduce the heliocentric flight-path angle at encounter with the next planet. The outbound swingby results in the lowest energy requirements for departure in 1977, 1979, 1983, and 1985. The inbound swingby results in the lowest energy for the 1981 opportunity.

The following is a typical sequence of the major events for an outbound Venus-swingby - Mars-orbiting mission. This sequence of events is the mission profile assumed for the swingby analysis; it is identical with the minimum-energy profile of reference 6 except that it swings by Venus and it does not allow a manned Mars landing. The manned landing is not attempted in this profile because it would require increased energy and longer stay time.

The sequence of events assumed for the analysis follows:

1. Earth orbit assembly occurs in a 262-n. mi. circular orbit. The completed assembly consists of three orbital launch stages and the space-craft.
2. Planetary injection is accomplished in three stages with an intermediate coast in an elliptical orbit between the stages. Injecting with three stages reduces the large gravity losses that would accompany a single burn of about 45 minutes.
3. Venus encounter occurs about 160 days after Earth departure. The Venus periapsis distance varies between 75 and 400 n. mi., depending on the particular trajectory.

4. Mars arrival occurs about 170 days after Venus passage. The periapsis altitude is 200 n. mi. The spacecraft is slowed to a 200- by 10 000-n. mi. orbit by the Mars orbit insertion (MOI) stage.

5. The planet is investigated during the 0- to 60-day stay; impact probes and soft landers are deployed.

6. Transearth injection (TEI) occurs at about 360 days into the mission.

7. Earth entry occurs after a total mission time of approximately 680 days; entry velocity is 50 000 fps or less.

For an inbound swingby, the position of the Venus encounter changes in the sequence.

Velocity Requirements

The velocity requirements for mission opportunity in 1977, 1979, 1981, 1983, and 1985 are summarized in figures 2, 3, and 4. This data is derived from reference 3 but includes a nominal velocity budget. The nonmission-dependent velocity budget is as follows:

1. Gravity and steering loss at Earth departure - 325 fps.
2. Earth-to-Venus midcourse corrections and artificial gravity spin-ups - 500 fps.
3. Venus-to-Mars midcourse corrections and artificial gravity spin-ups - 500 fps.
4. Gravity and steering loss at MOI - 100 fps.
5. Mars orbit maneuver and artificial gravity spin-ups - 500 fps.
6. Gravity and steering loss for TEI - 50 fps.
7. Mars-to-Earth midcourse correction and artificial gravity spin-ups - 500 fps.

The total velocity budget for the spacecraft is 2150 fps, which is 500 fps higher than that required for the minimum-energy mission. The increase is caused by the increased midcourse requirements for an accurate Venus passage.

Figure 2 shows the velocity requirements for a 12-hour stay time at Mars. This does not imply a free-return type of trajectory but rather a retrograde maneuver to orbit, a coast for one revolution, and an immediate posigrade burn for TEI. The data is included here to provide continuity from 0- to 60-day stays.

The vertical bars in figures 2, 3, and 4 indicate the requirements for a 20-day Earth departure window assuming TMI occurs on time. The velocity requirements for the minimum-energy mission are shown in figure 5 with the same velocity budget (except for the additional 500-fps midcourse ΔV) included in the data as in the swingby case. The maximum velocity required onboard the spacecraft (MOI and TEI) occurs in 1983 for both the Venus swingby and the minimum-energy mission. The Venus swingby requires about 11 000 fps more velocity onboard the spacecraft in 1983 than is required with the minimum-energy mission. However, the Venus swingby in 1979 and 1985 can be accomplished with an onboard capability of 16 000 fps, which is only 6000 fps higher than the 1985 minimum-energy mission. The 1979 and 1985 mission durations are 680 days with the Venus swingby and 1000 days with the minimum-energy profile.

Spacecraft Weights

The total spacecraft weight required for the 1977 through 1985 missions is plotted in figures 6 through 10 as a function of payload deployed in Mars orbit and for orbiting times from 0 to 60 days. The total spacecraft weight is that weight required immediately following transplanetary injection. The basic module weights are listed below:

Mission module (dry, crew 4), lb	35 000
Meteoroid protection, lb	5 740
Earth entry module, lb	15 100
Gravity tube, lb	3 000
Expendables, lb.	14 000
Experiments (onboard), lb.	<u>1 000</u>
Total weight, lb	73 840

The specific impulse and stage mass fraction for the MOI and TEI stages are 400 seconds and 20 percent.

A booster capability of 400 000 lb to a 262-n. mi. orbit would result in about 380 000 lb of useful spacecraft payload where 20 000 lb is lost due to docking structures, spacecraft/booster interstage, rendezvous propellant. The orbital launch vehicle (OLV) useful weight would be about 20 000 lb less due to propellant boiloff, (i.e., useful weight 360 000 lb). If the booster capability were increased to 500 000 lb to orbit, then the useful spacecraft weight is 480 000 lb and the OLV weight is 460 000 lb.

The spacecraft can be launched in one or more pieces and thus exceed 380 000 lb or 480 000 lb, but a single launch is preferable if possible.

The velocity available for transplanetary injection is listed in table I as a function of the number of OLV stages. It is assumed that the spacecraft weight is 20 000 lb less than the booster payload capability and the OLV useful weight is 40 000 lb less than the booster capability. The OLV specific impulse is 433 seconds, and the stage mass fraction is 10 percent. Three OLV's will provide sufficient injection velocity for all five opportunities shown in figures 2 through 4; in 1981 only two OLV's are needed. However, this assumes that the total spacecraft weight is equal to or less than the single launch capability. The 1977, 1981, and 1983 opportunities require a total spacecraft weight greater than 480 000 lb so that even the larger booster is not enough with the single-launch spacecraft concept. The 1979 and 1985 opportunities require lesser spacecraft weights and are feasible missions, at least from performance considerations.

Table II is a weight breakdown of a 380 000-lb spacecraft capable of performing the 1979 mission with a 10 000-lb payload and a 30-day Mars stay, or the 1985 mission with a 5000-lb payload and a 10-day stay in Mars orbit. If the spacecraft weight is increased to 480 000 lb, then the payload for the 1979 mission is 50 000 lb and the 1985 mission payload is 30 000 lb with a 25-day stay. An increase in spacecraft weight could be used for fuel instead of payload and a 480 000-lb spacecraft with 10 000-lb payload is capable of 18 230 fps (i.e., MOI ΔV and TEI ΔV). This would make the 1977 and 1981 departures marginally available (i.e., the 20-day window be reduced).

CONCLUSIONS

If used for Mars orbiting missions during the 1977 through 1985 cycle, the Venus swingby technique reduces the total mission duration from the 2.8 years for the minimum-energy technique to 1.8 years. The reduction is achieved at the cost of the following items when compared with the minimum-energy mission:

1. An 80 percent reduction in Mars orbit staytime.
2. A 90 percent reduction in Mars orbit payload.
3. A 25 percent increase in Earth entry velocity.
4. A 60 percent reduction in Earth departure windows.

TABLE I.- TRANS-MARS INJECTION VELOCITY CAPABILITY^a

Number of OLV's	Single OLV weighting 380 000 lb	Single OLV weighting 480 000 lb
	ΔV, fps	ΔV, fps
1	7 897	7 989
2	12 690	12 820
3	16 140	16 291
4	18 836	19 002
5	21 050	21 226

^aThe spacecraft weight equals that of one OLV.

TABLE II.- SPACECRAFT WEIGHTS FOR A VENUS-FLYBY - MARS-ORBITAL
MISSION IN 1979 WITH A 30-DAY STAY AT MARS

Component	Weight, lb	Total weight, lb
MM dry (crew - 4)	35 000	
Meteoroid protection.	5 740	
EEM	15 100	
Gravity tube.	3 000	
Onboard experiments	1 000	
Expendables	14 000	
Experiments (deployed in orbit)	<u>10 000</u>	
Total		83 840
MOI stage (dry)	38 308	
TEI stage (dry)	<u>11 052</u>	
Total		49 360
Total dry weight.		133 200
MOI fuel.		191 537
TEI fuel.		<u>55 263</u>
GRAND TOTAL		380 000

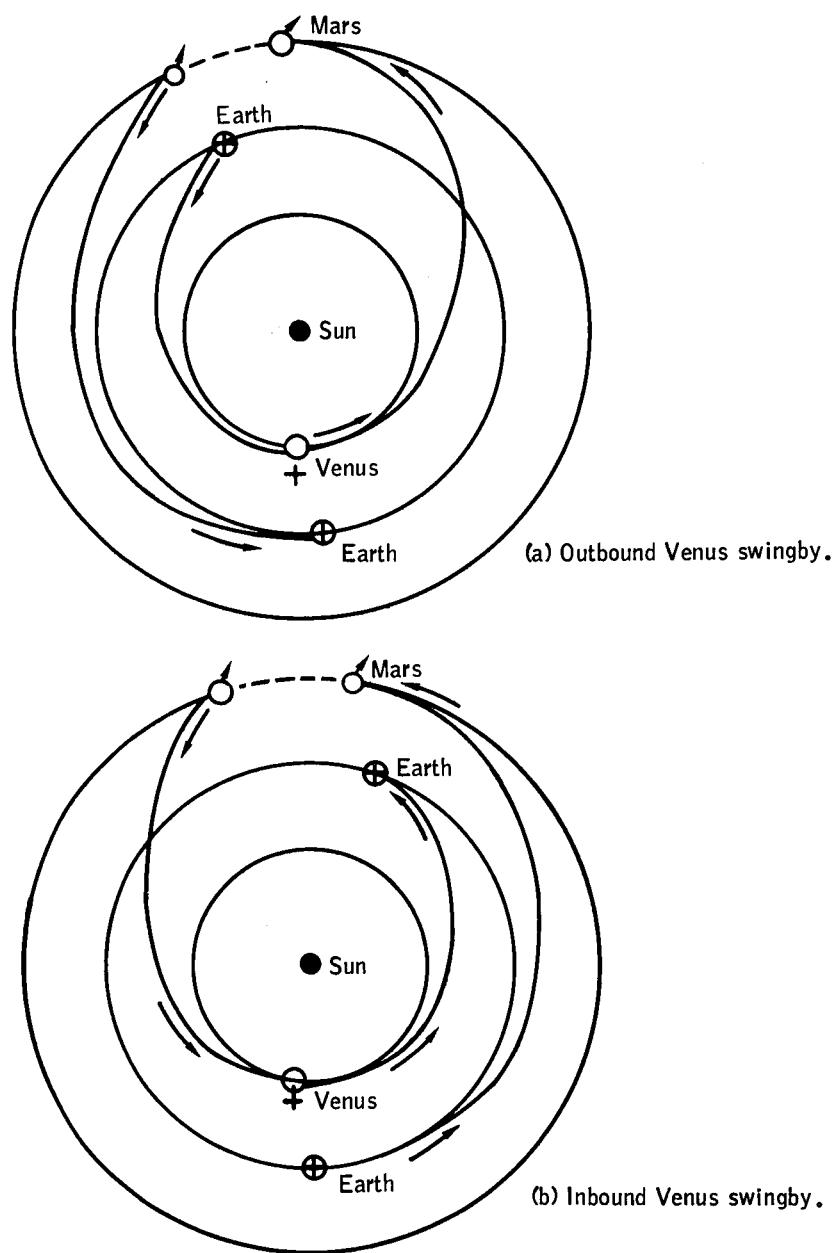


Figure 1.- A schematic of the Venus swingby technique for Mars orbiting missions.

CHANGE SHEET
FOR
MSC INTERNAL NOTE 67-FM-82, DATED JUNE 28, 1967
AS-503A/AS-504A REQUIREMENTS FOR THE RTCC:
POWERED-FLIGHT SIMULATIONS

By Ernest M. Fridge, OMAB, and Jerome D. Yencharis, LMAB

Change 2
April 23, 1968

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Page 1 of 19
(with enclosures)

After the following enclosures, which are replacements, have been inserted, insert this CHANGE SHEET between the cover and title page and write on the cover "CHANGE 2 inserted".

NOTE: A black bar in the margin indicates the area of change.

1. Replace pages 83 through 87.
2. Replace pages 93 through 96.
3. Replace pages 100, 102, 114, and 115.
4. Replace pages 120 through 123.

Change 2, April 23, 1968.

SYMBOLS FOR SUBROUTINE HYPER

INPUT

Constants

μ gravitational constant of earth

GE gravitational acceleration

Constant Input - Defined When Launch Day is Defined

T_{st} coast time in earth parking orbit before restart test is entered for first injection opportunity

T_{st2} additive to T_{st} for coast time criteria for second injection opportunity

T_{st3} additive to T_{st} for coast time criteria for third injection opportunity

β, β_2, β_3 central angle from radial at restart preparation initiation to node of parking orbit plane and desired cutoff plane, for first, second, third injection opportunities, respectively

(a) T_{lo} reference time of launch; usually time of launch for first opportunity in window, measured from midnight

$(\cos \sigma)_n$ arc radius of perigee circle

$(c_3)_n$ desired cutoff energy; twice vis viva energy

$(eN)_n$ desired eccentricity at cutoff

$(RA, DEC)_n$ right ascension, declination of target vector

$(t_D)_n$ difference in time of launch from reference time of launch ($n = 1, 2, 3, \dots, 15$; $\cos \sigma, c_3, e_N$, RA, DEC are stored functions of TT)

^a Input if TUL = 1.

Change 2, April 23, 1968

α_{TS}^*	nominal central angle between the target vector, \bar{T} and the nodal vector \bar{S}
$K_{\alpha 1}, K_{\alpha 2}$	coefficients of α_{TS} polynomial
(a) θ_{eo}	right ascension of launch site at reference time of launch (T_{LO})
(a) ϕ_L	geodetic latitude of launch site
(a) DTGM	constant Δ time from TB6 to time at which IGM is entered
(a) DTIG	constant Δ time from TB6 to ignition
(a) K_{po}, K_{pl}, K_{p2}	coefficient of pitch, yaw polynomials in restart guidance
(a) K_{yo}, K_{yl}, K_{y2}	
h_{1n}, h_{2n}, h_{3n} ($n = 0, 1, 2, 3, 4$)	coefficients for launch azimuth polynomials
t_{D1}, t_{SD1} t_{D2}, t_{SD2} t_{D3}, t_{SD3}	
t_{DS0}, t_{DS1}' t_{DS2}, t_{DS3}'	values of t_D indicating bounds for usage of each of the three AZ polynomials
	Real-time Inputs
I_{OPP}	indicates injection opportunity; if = 1, first opportunity = 2, second opportunity = 3, third opportunity
t_c	time of coast in earth parking orbit

^a Input if TUL = 1.

Change 2, April 23, 1968.

(a) T_L	actual time of launch; usually measured from midnight
(a) AZ	launch azimuth
(a) T_{4A}	actual burn time for first S-IVB burn
(a) T_{4N}	nominal burn time for first S-IVB burn
(a) $\Delta T'_{4M}$	limit on difference for first S-IVB burn time
R_N	nominal radius at ignition
f	estimate of true anomaly at cutoff
(a) T'_3	estimate of burn time for third IGM stage
(a) K_{t3}	coefficient used to determine T'_3
(a) τ_3	estimate of burn time for vehicle depletion during third IGM stage
(a) Δt	computation interval
(a) \bar{R}	current position vector
(a) \bar{V}	current velocity vector
(a) R	$ \bar{R} $
TUL	indicates if target update has occurred; if = 0, no update = 1, update targeting and input the following
e	desired eccentricity of cutoff ellipse
C_3	(see above)
i, θ_N	(see below)
α_D	true anomaly of descending node of desired cutoff plane
f	estimate of true anomaly at cutoff
TB6	time to initiate restart preparation and the parameters indicated by an asterisk (a) above

^aInput if TUL = 1.

Change 2, April 23, 1968

INTERMEDIATE PARAMETERS

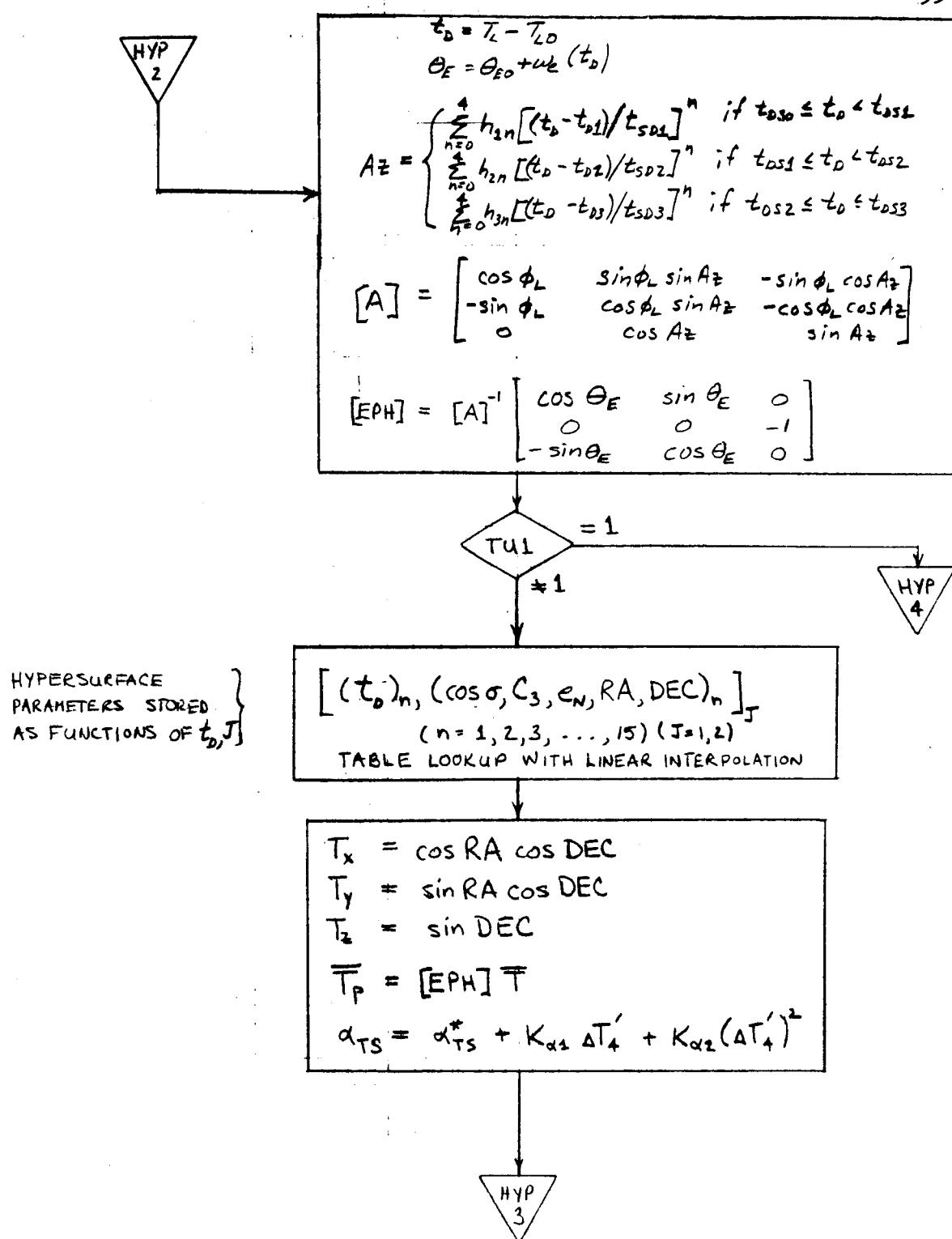
\bar{s}	nodal vector; defines node of desired cutoff plane and parking orbit plane at time of restart preparation initiation
$x_p \}$	pitch, yaw in restart guidance
$xy \}$	
[GG]	transformation matrix, from plumbline to nodal coordinate system; nodal coordinates referenced to parking orbit plane and equatorial plane
\dot{w}	weight flow rate
w	current vehicle weight
(F/m)	thrust acceleration magnitude
\bar{s}_T	thrust acceleration vector

OUTPUT

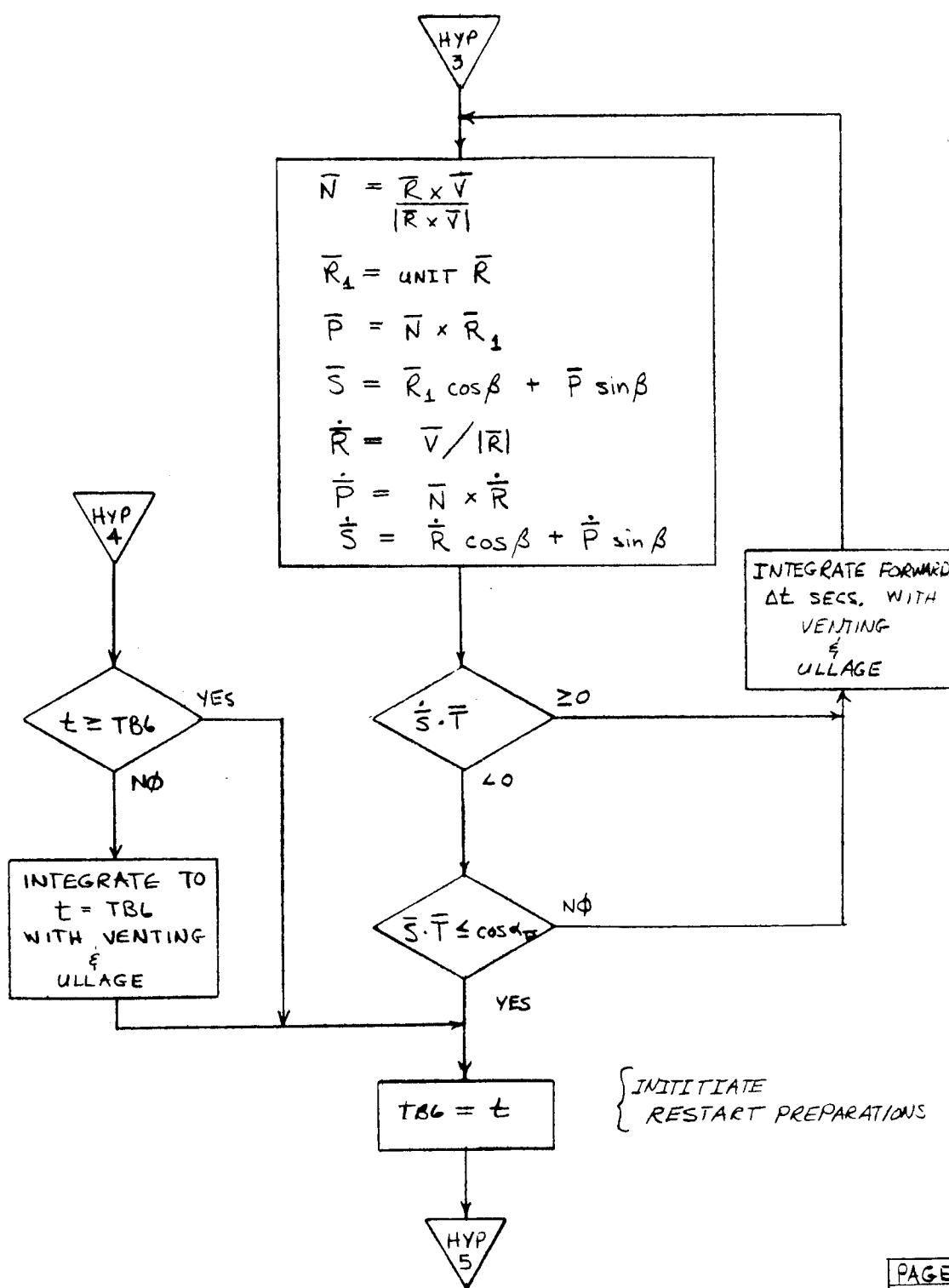
TB6	time of restart preparation initiation
T_{IGM}	time at which IGM is entered
t_{IGN}	time of ignition
[G]	transformation matrix; plumbline to nodal coordinates; nodal coordinate system referenced to desired cutoff plane and equatorial plane
R_T	estimated radial magnitude at cutoff
V_T	estimated velocity magnitude at cutoff
γ_T	estimated flight-path angle at cutoff
G_T	gravitational acceleration at cutoff

Change 2, April 23, 1968

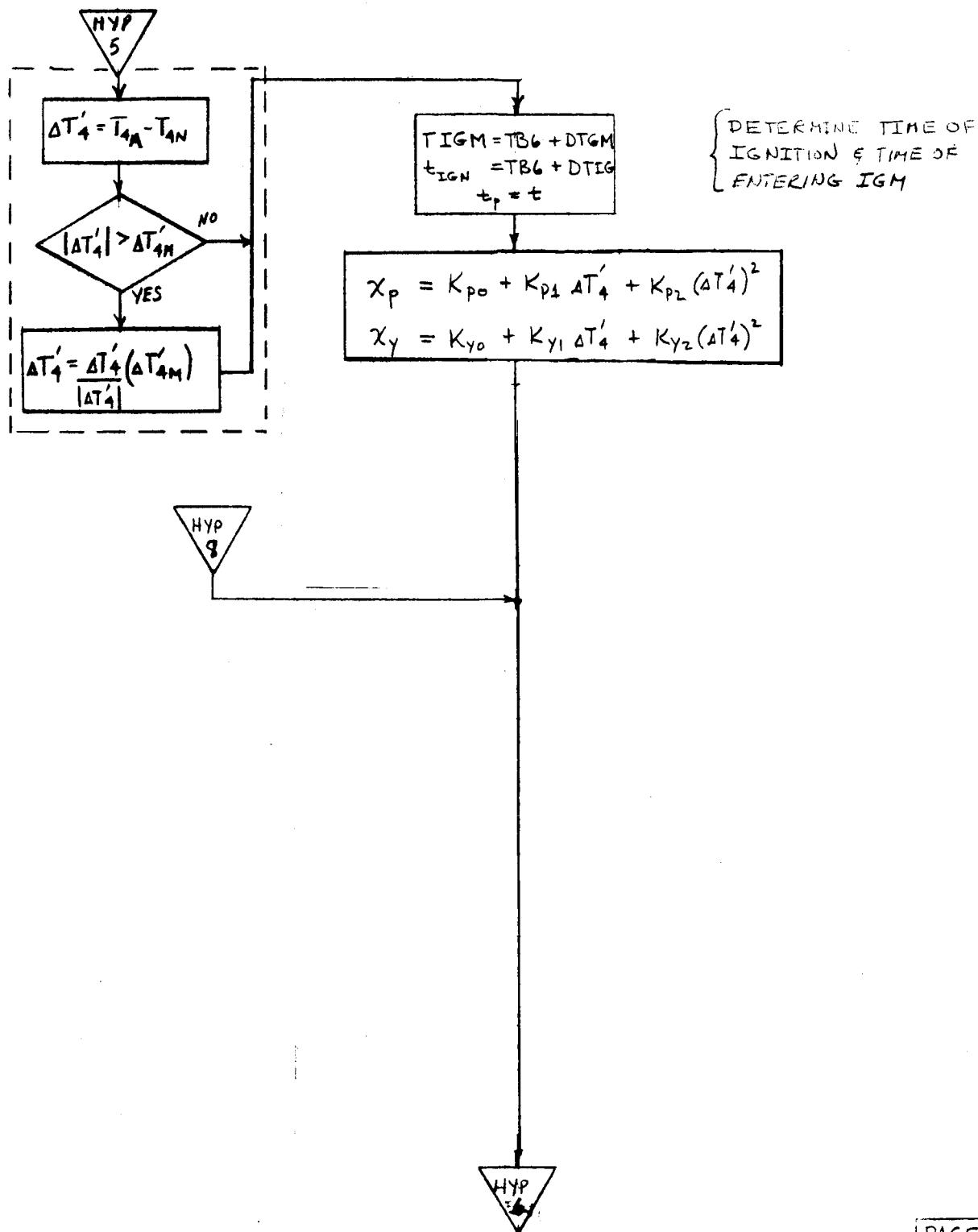
T_3	estimated burn time for third IGM stage
τ_3	estimated burn time for vehicle depletion after mixture ratio shift
[EPH]	transformation matrix; ephemeral to plumbline
$[\bar{r}, \bar{v}, t]$	state at $t = t_{ign} + DTGM$
i	inclination of desired cutoff ellipse
θ_n	descending node of desired cutoff ellipse, measured as defined above
P_n	semilatus rectum of nominal cutoff ellipse
e	eccentricity of desired cutoff ellipse
α_D	true anomoly of descending node
f	estimated true anomoly at cutoff
p	semilatus rectum of desired cutoff ellipse



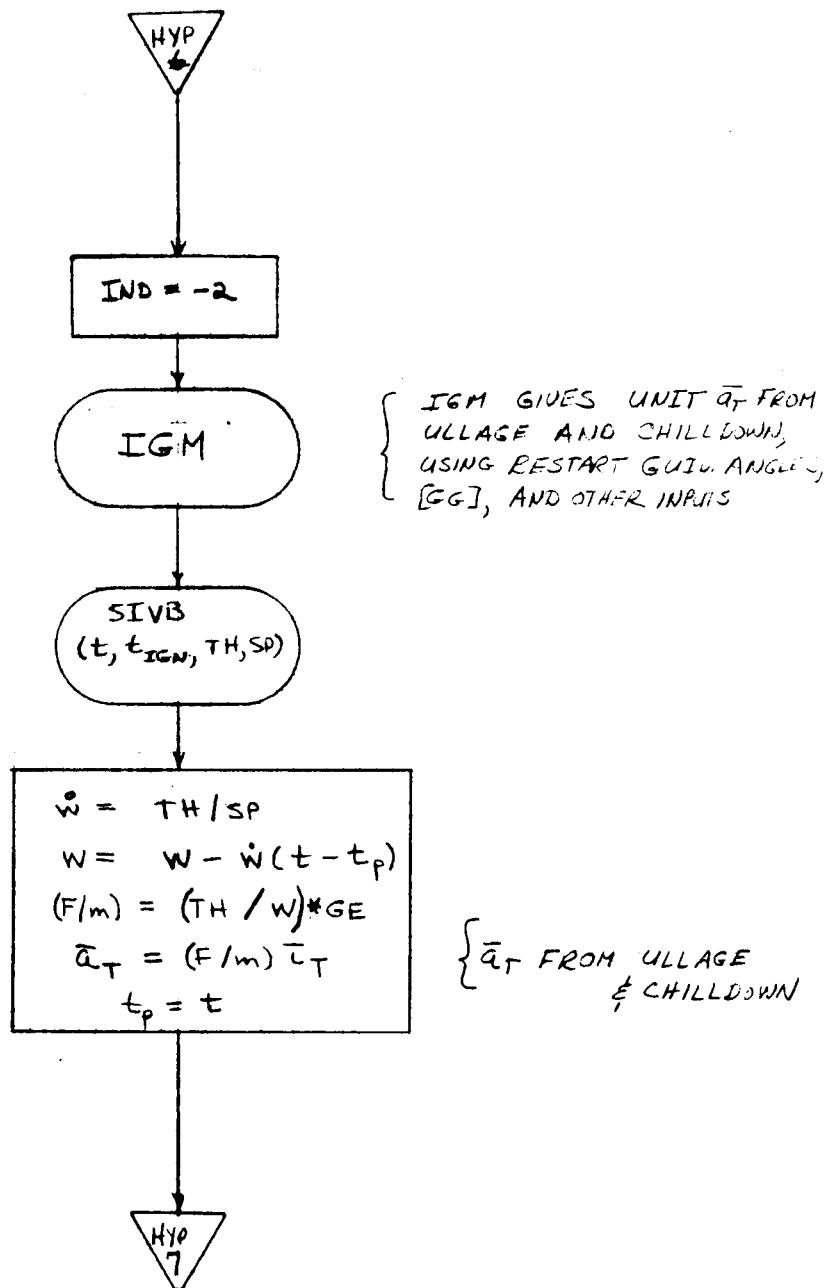
Flow chart B-2.- Detailed flow for subroutine HYPER. - Continued



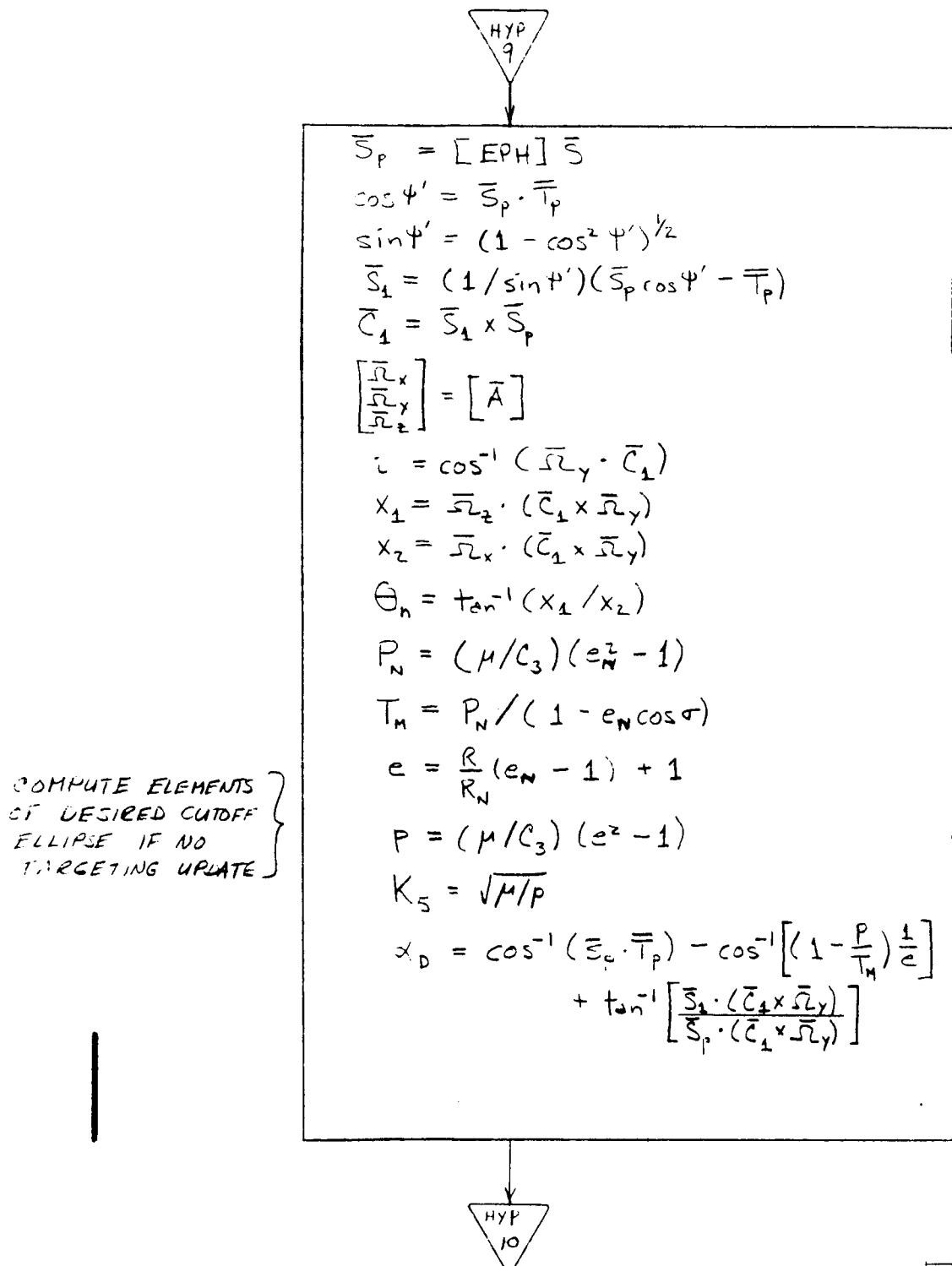
Flow chart B-2.- Detailed flow for subroutine HYPER. - Continued



Flow chart B-2.- Detailed flow for subroutine HYPER. - Continued

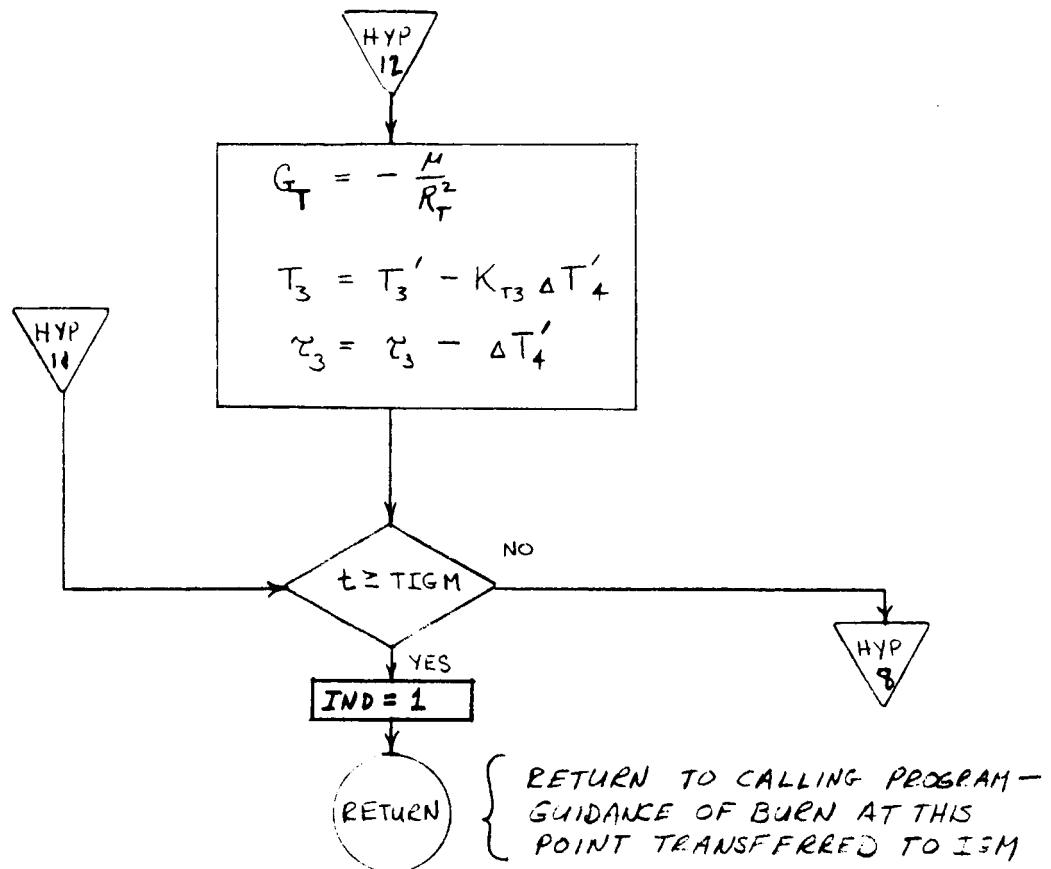


Flow chart B-2.- Detailed flow for subroutine HYPER. - Continued



Flow chart B-2.- Detailed flow for subroutine HYPER. - Continued

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PAGE	OF
9	9

Flow chart B-2.- Detailed flow for subroutine HYPER. - Concluded

Change 2, April 23, 1968.

 K_5 function of μ , p

IND

flag; if

= -1 indicates restart guidance angles
to be transformed to thrust acceleration
vector= +1 indicates $t = (t_{ign} + DTGM)$

$$\begin{bmatrix} R_T, V_T, \gamma_T, G_T \end{bmatrix}$$

(see below for definition)

Initialized Data - Constant

 V_1
 $= 0 \quad \left. \begin{array}{l} \\ \end{array} \right\} \text{cutoff logic parameters}$
 V_2
 $= 0 \quad \left. \begin{array}{l} \\ \end{array} \right\} \text{cutoff logic parameters}$
 $\Delta t'_2$
 $= T_3 + T_2 \quad \left. \begin{array}{l} \\ \end{array} \right\}$
 T_{go}
 $= 0 \quad (\text{see below})$
 t_p

IMRS

flag initially set to -1; if
 = -1 indicates mixture ratio shift has
 not occurred
 = +1 indicates MRS has occurred, but
 guidance is in artificial tau mode
 = 0 indicates completion of artificial
 tau mode

PC

 $= 0 \quad (\text{see below})$

UP

flag initially set to -1; if
 = -1 indicates update of burn time
 estimate and recycle through range angle
 computations
 = +1 indicates no update of burn time
 estimate

Variables

 T_2

second (guidance) stage burn time

 V_{ex2}

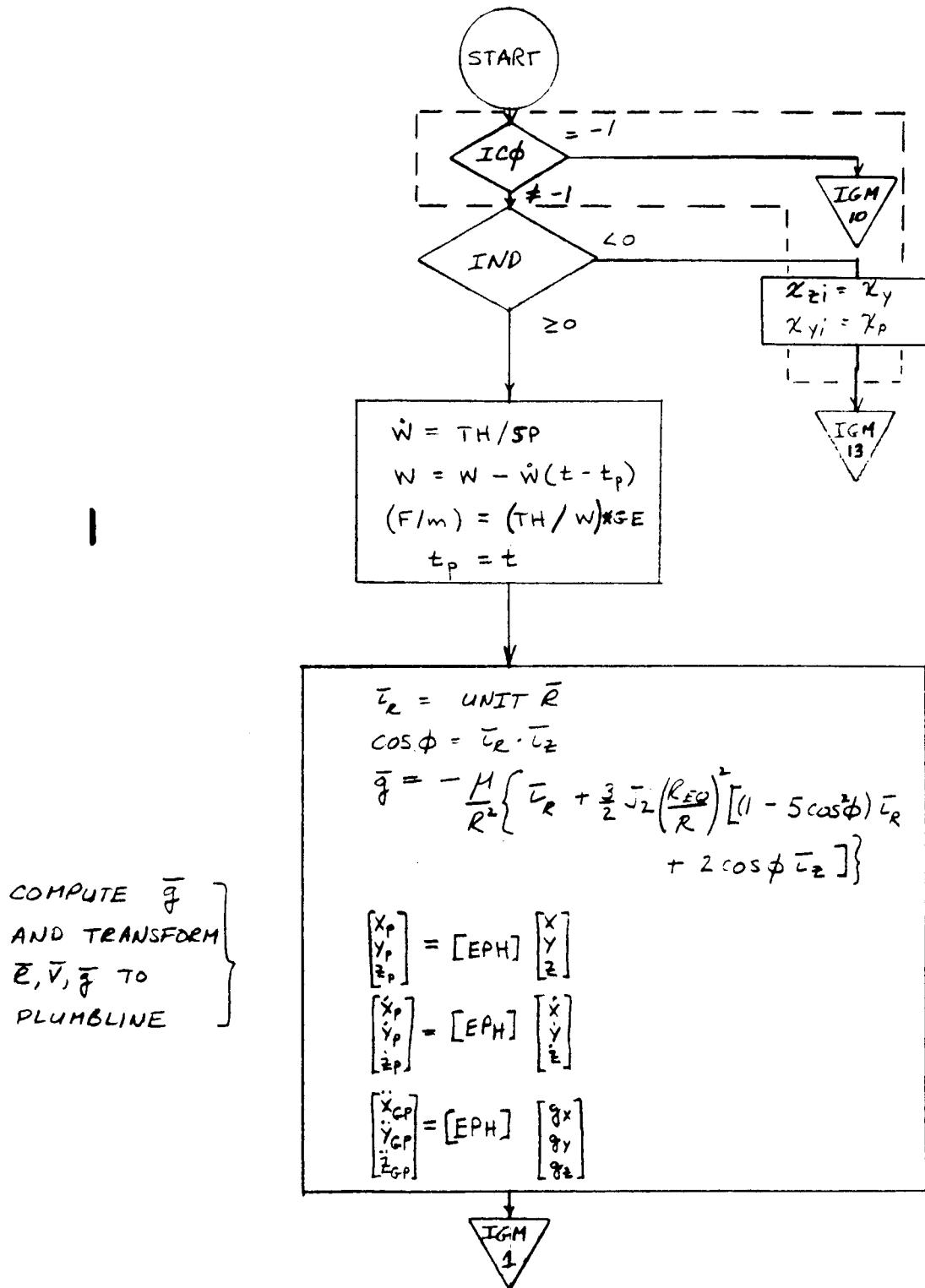
exhaust velocity - second IGM stage

$$\begin{array}{l} T_{2N} \\ C_o \end{array}$$

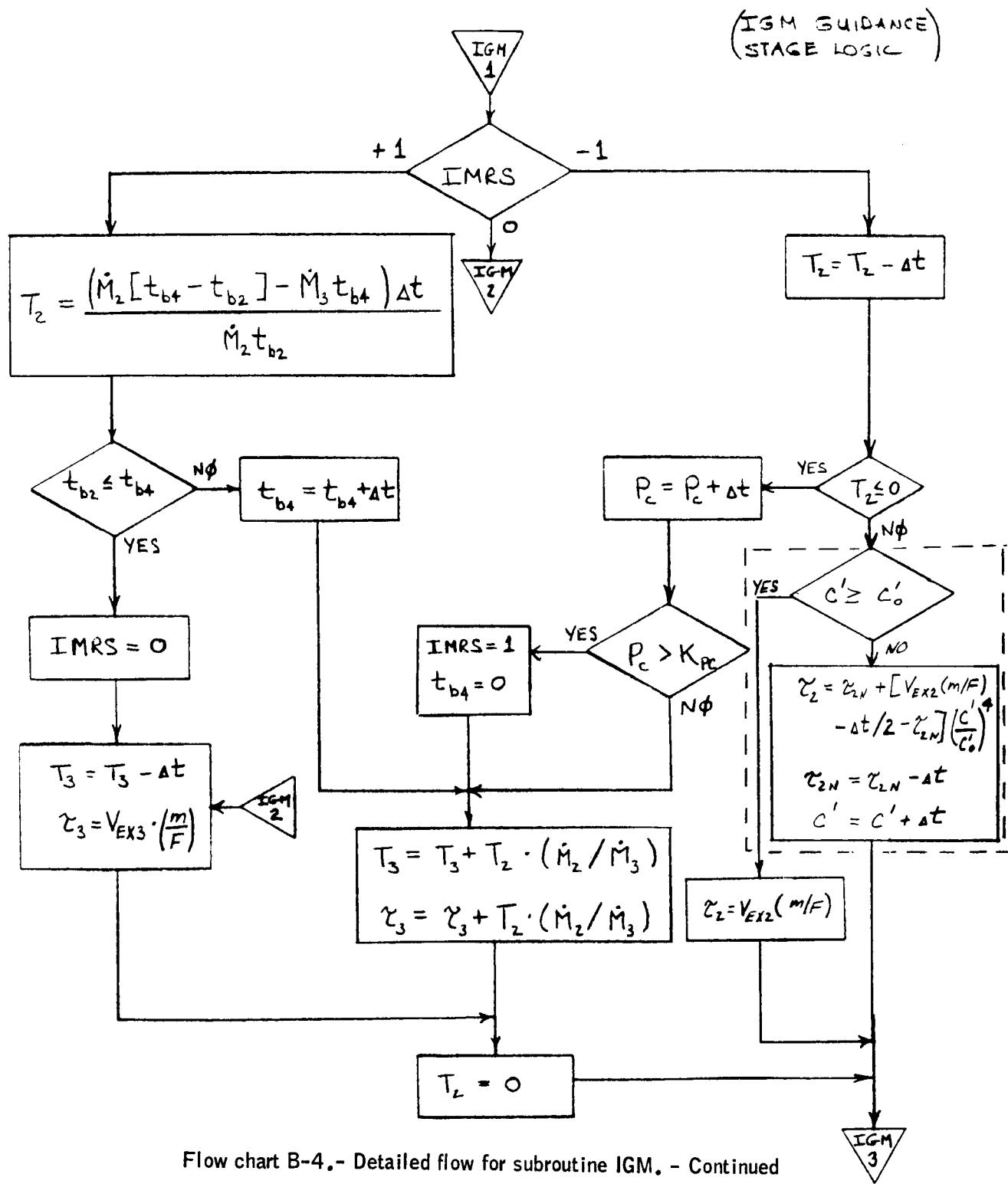
variables used in artificial tau mode

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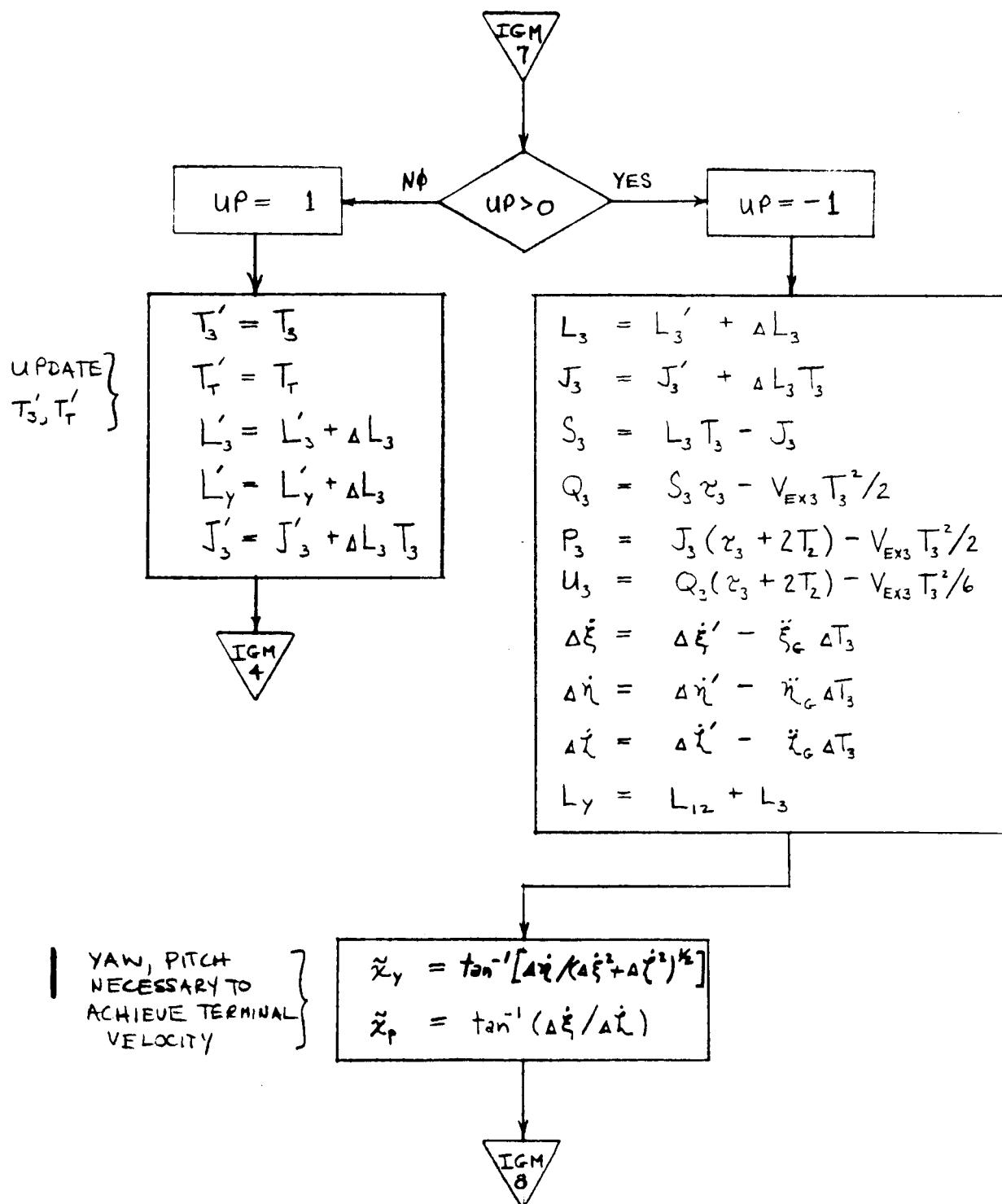
DETAILED FLOW
SUBROUTINE IGM



Flow chart B-4.- Detailed flow for subroutine IGM.

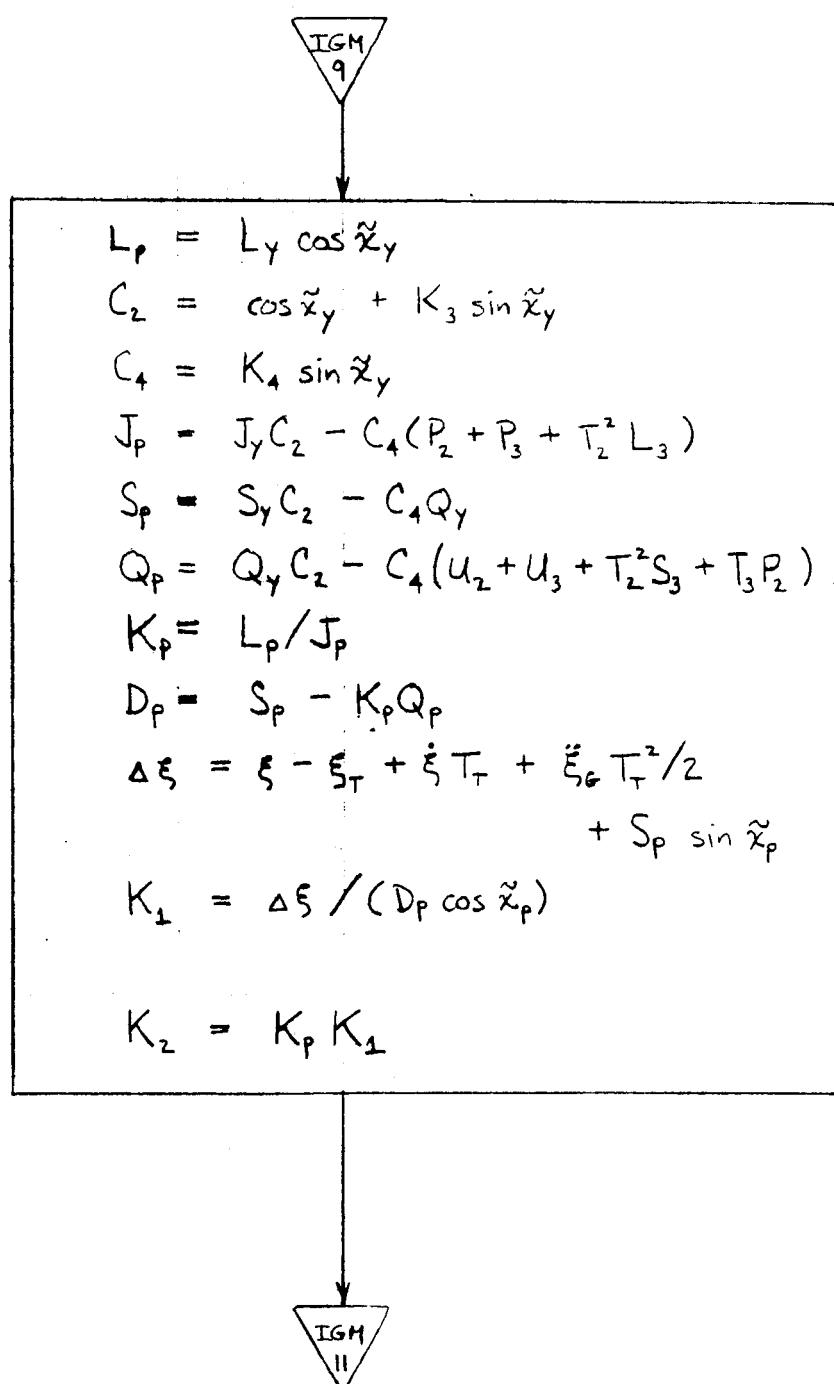


Flow chart B-4.- Detailed flow for subroutine IGM. - Continued

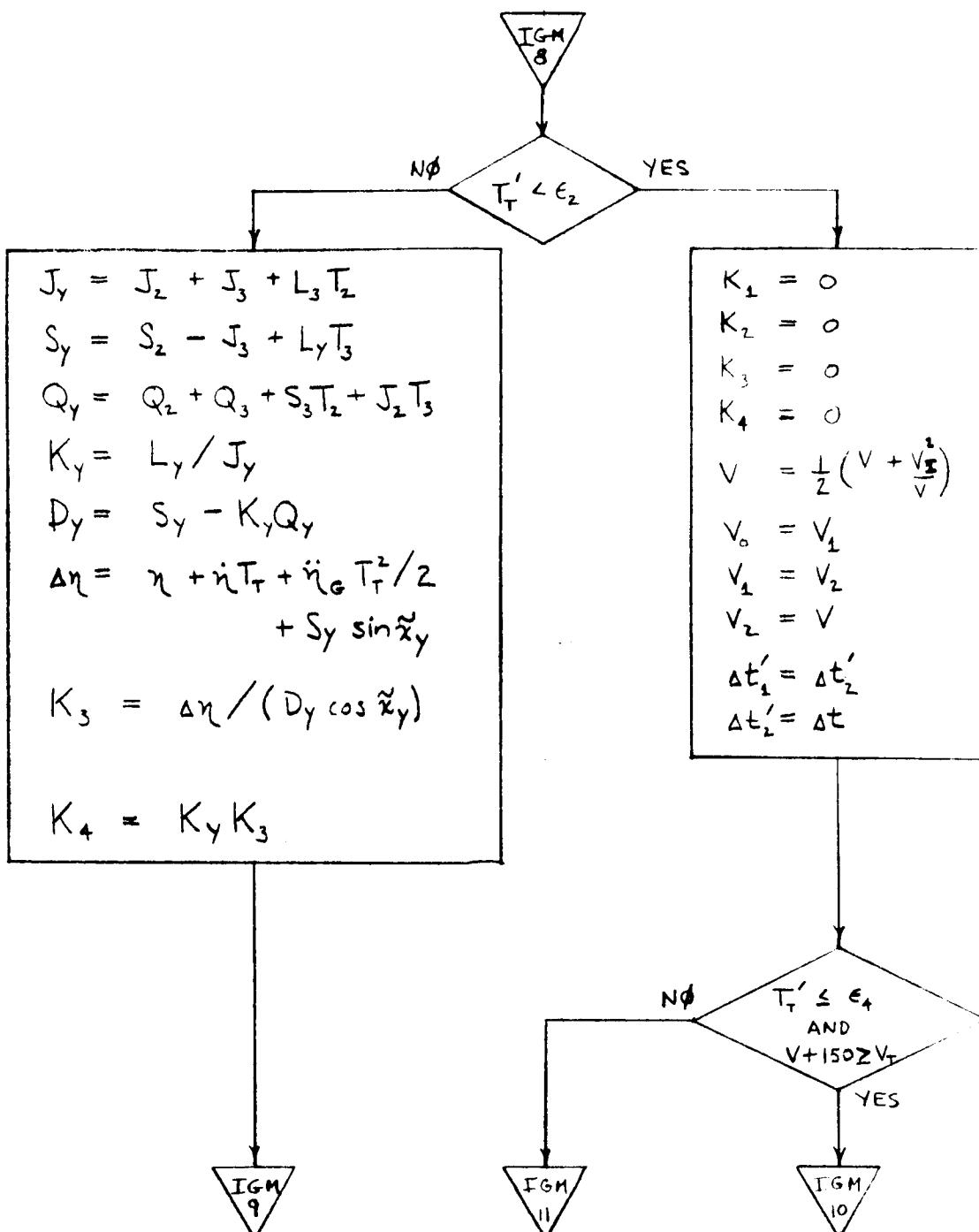


Flow chart B-4.- Detailed flow for subroutine IGM. - Continued

Change 2, April 23, 1968

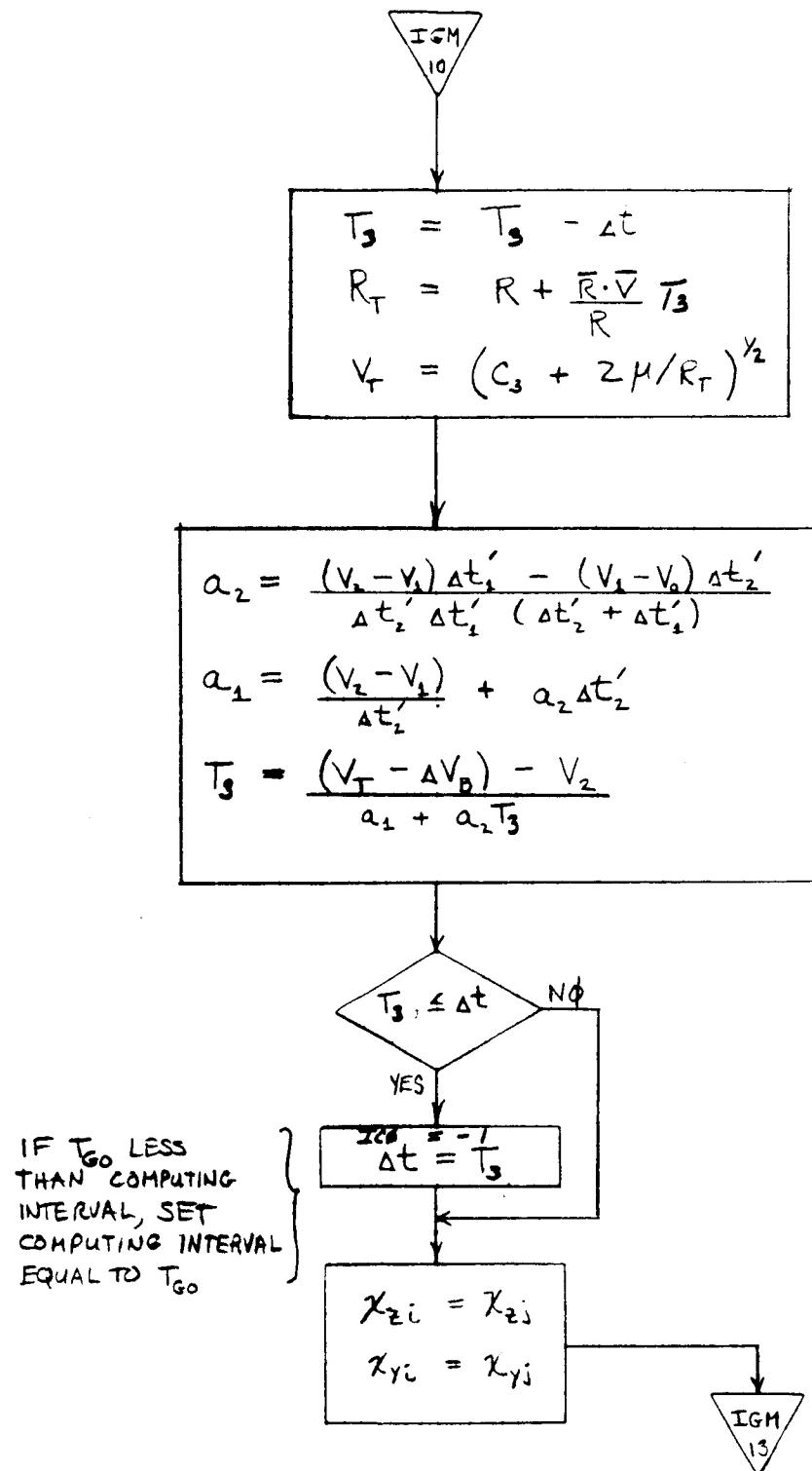


Flow chart B-4.- Detailed flow for subroutine IGM.- Continued



Flow chart B-4.- Detailed flow for subroutine IGM. - Continued

(CUTOFF LOGIC)



Flow chart B-4.- Detailed flow for subroutine IGM. - Continued

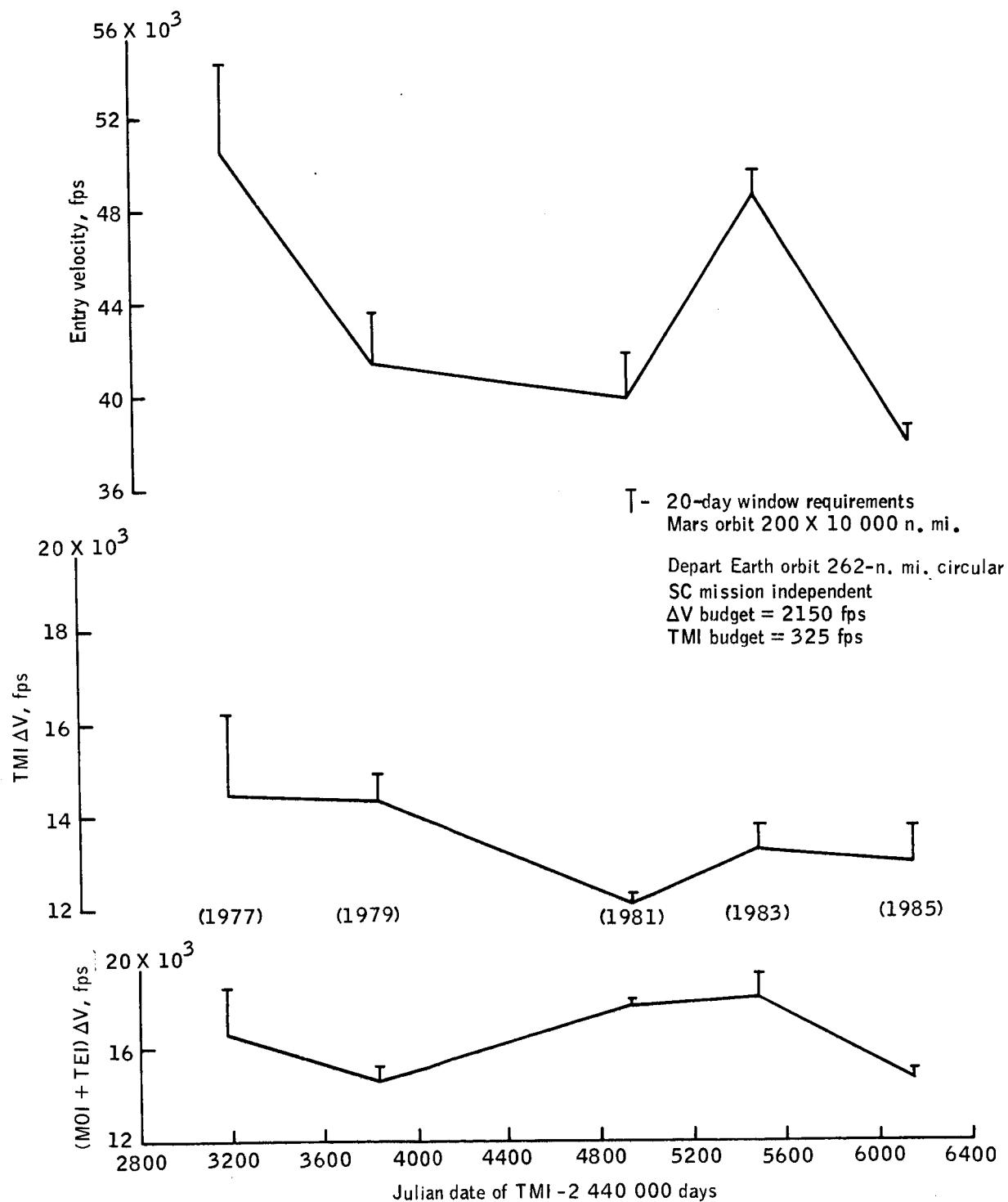


Figure 2.- Venus swingby mission velocity requirements with 12-hours stay time at Mars.

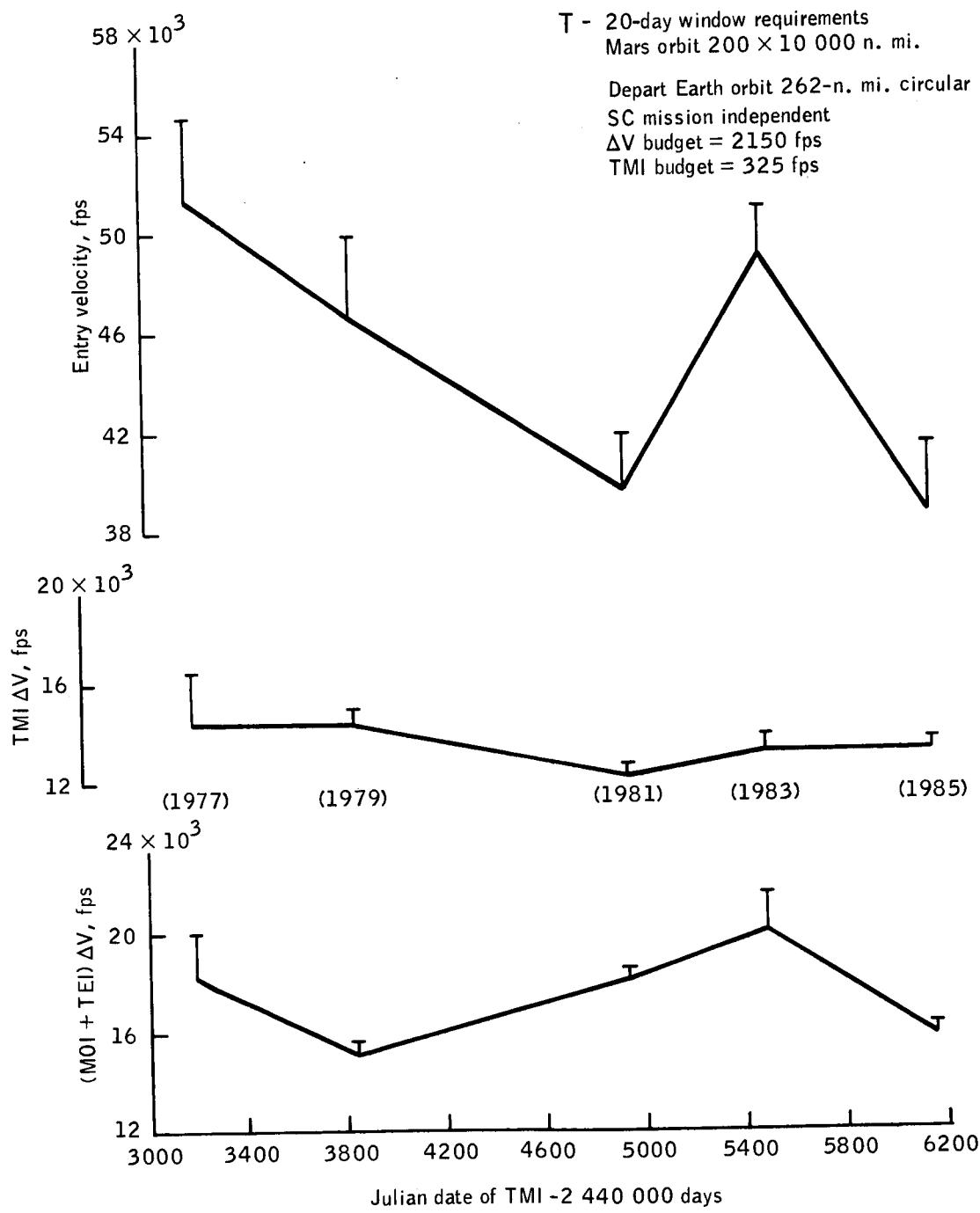


Figure 3.- Venus swingby mission velocity requirements with 30-day stay time at Mars.

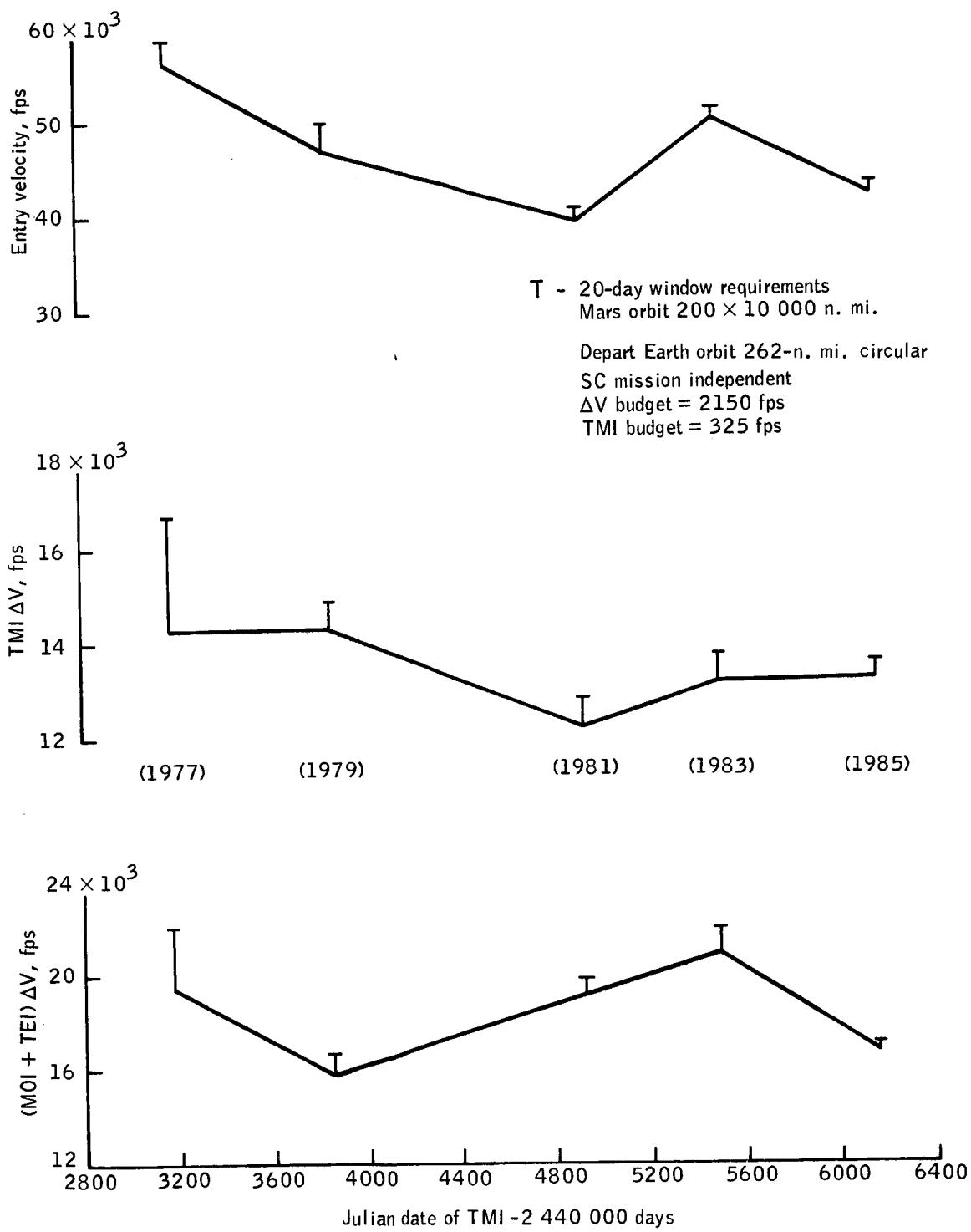


Figure 4. - Venus swingby mission velocity requirements with 60-day stay time at Mars.

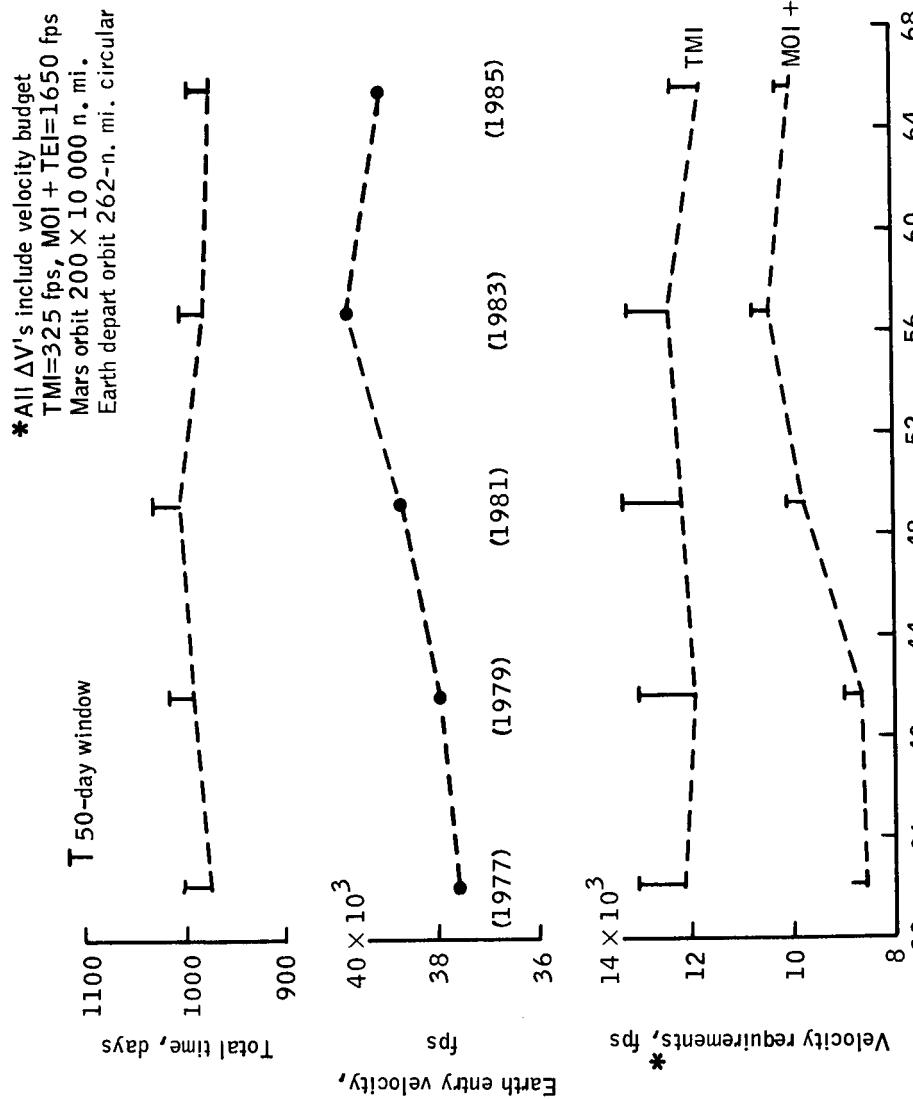


Figure 5.- Minimum energy mission velocity requirement with Martian stay times of 300 to 465 days.

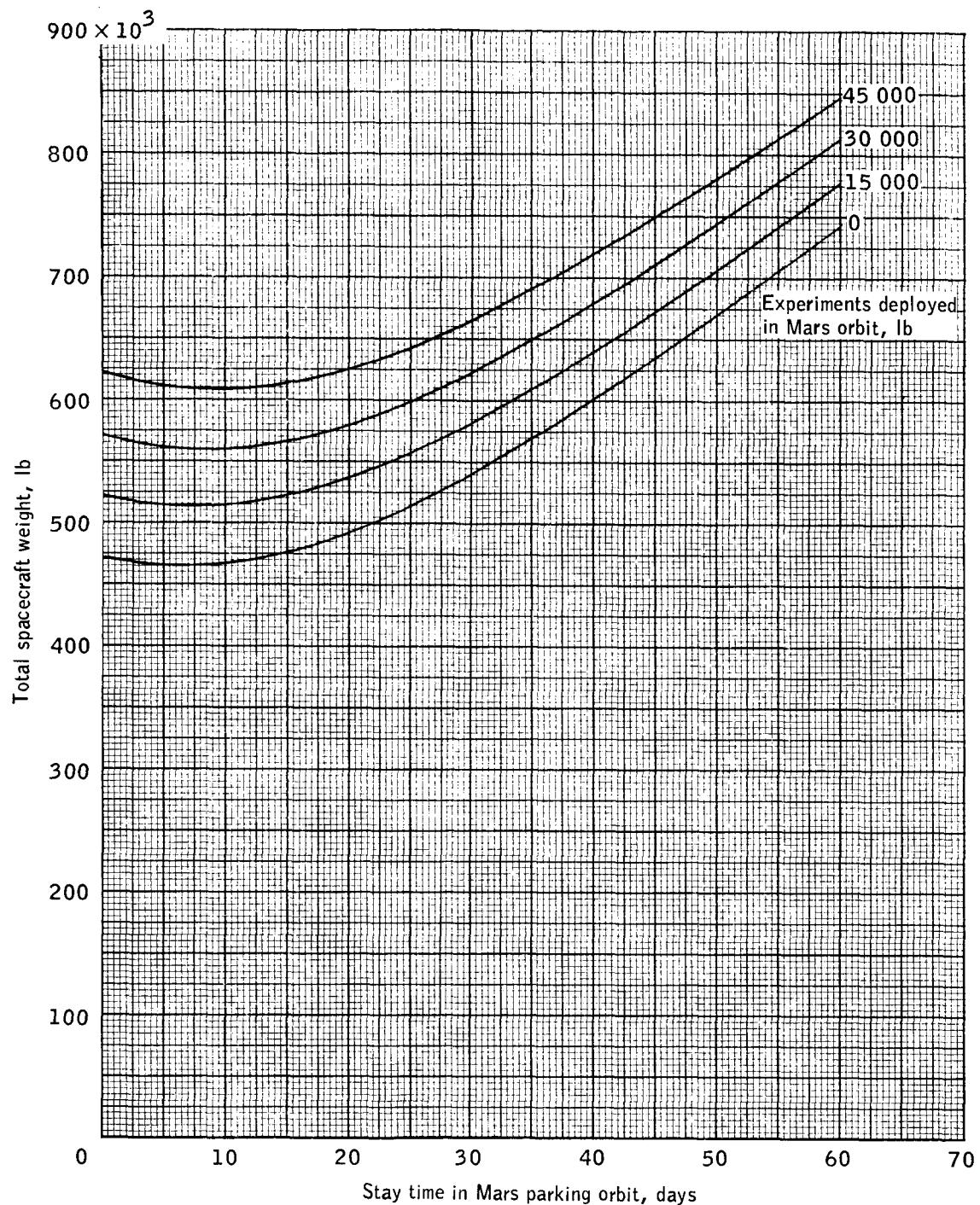


Figure 6.- 1977 Mars orbiting mission using Venus outbound swingby. Total mission duration is 600 to 640 days.

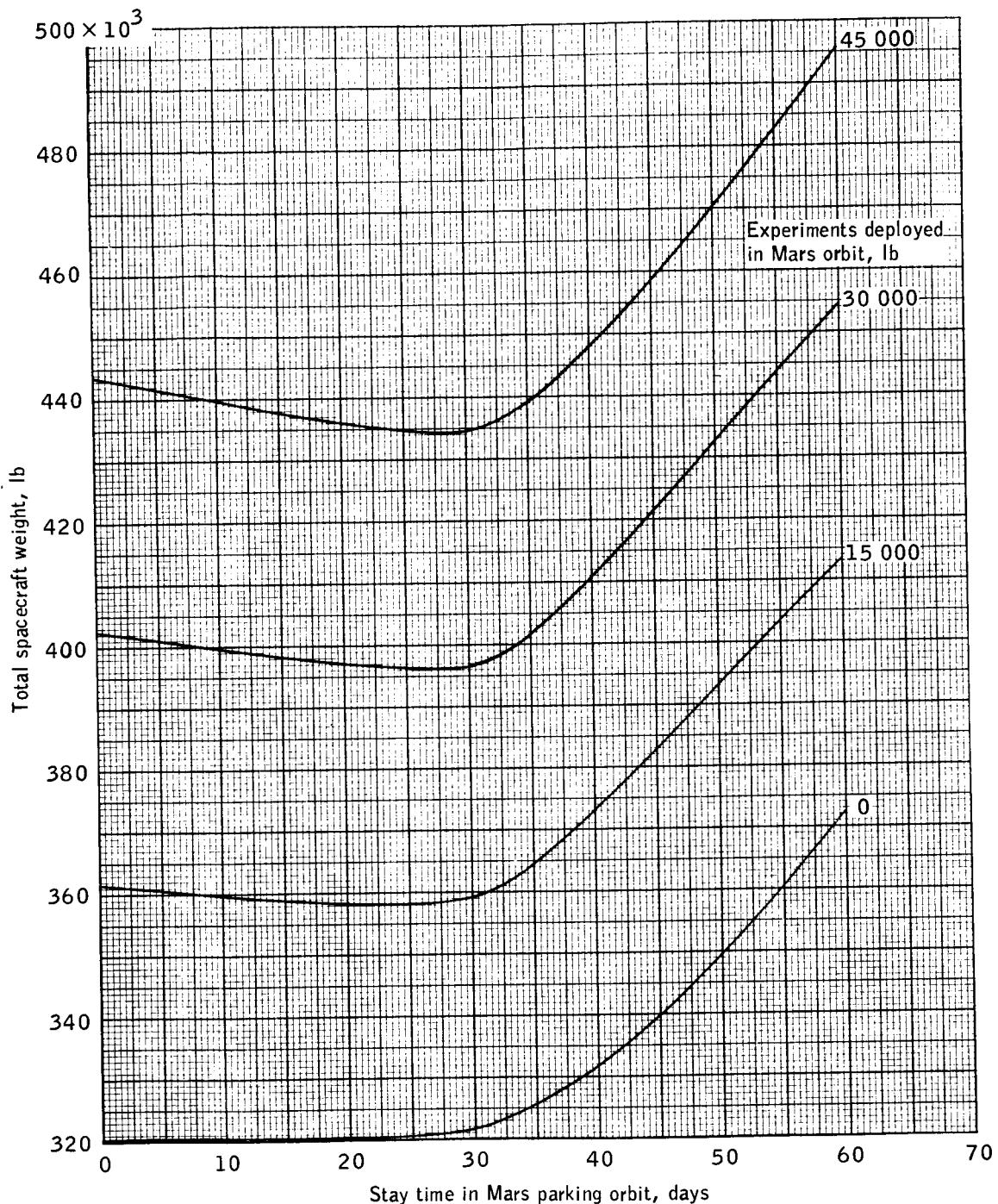


Figure 7.- 1979 Mars orbiting mission using Venus outbound swingby. Total mission duration is 640 to 680 days.

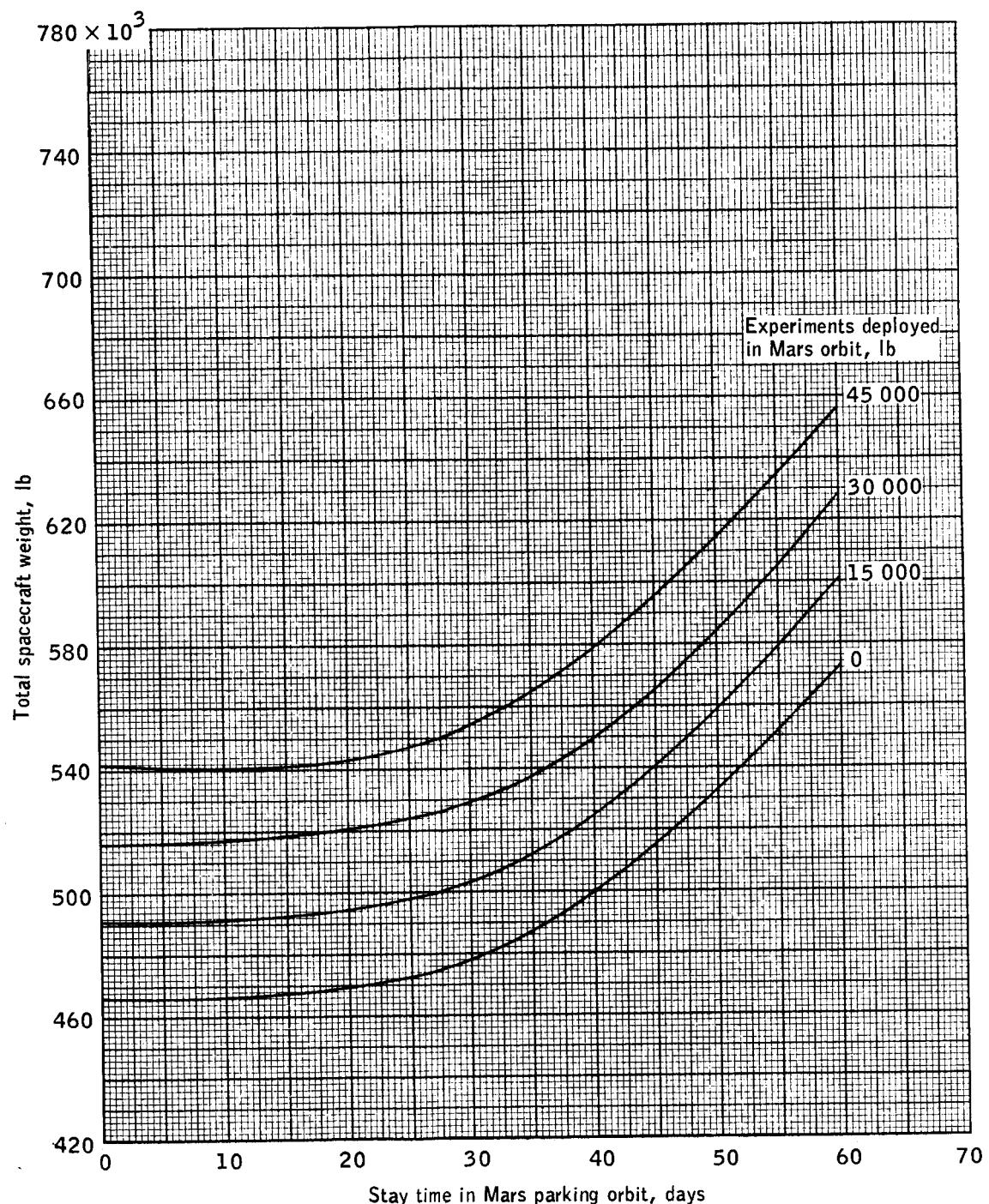


Figure 8.- 1981 Mars orbiting mission using Venus inbound swingby. Total mission duration is 600 to 640 days.

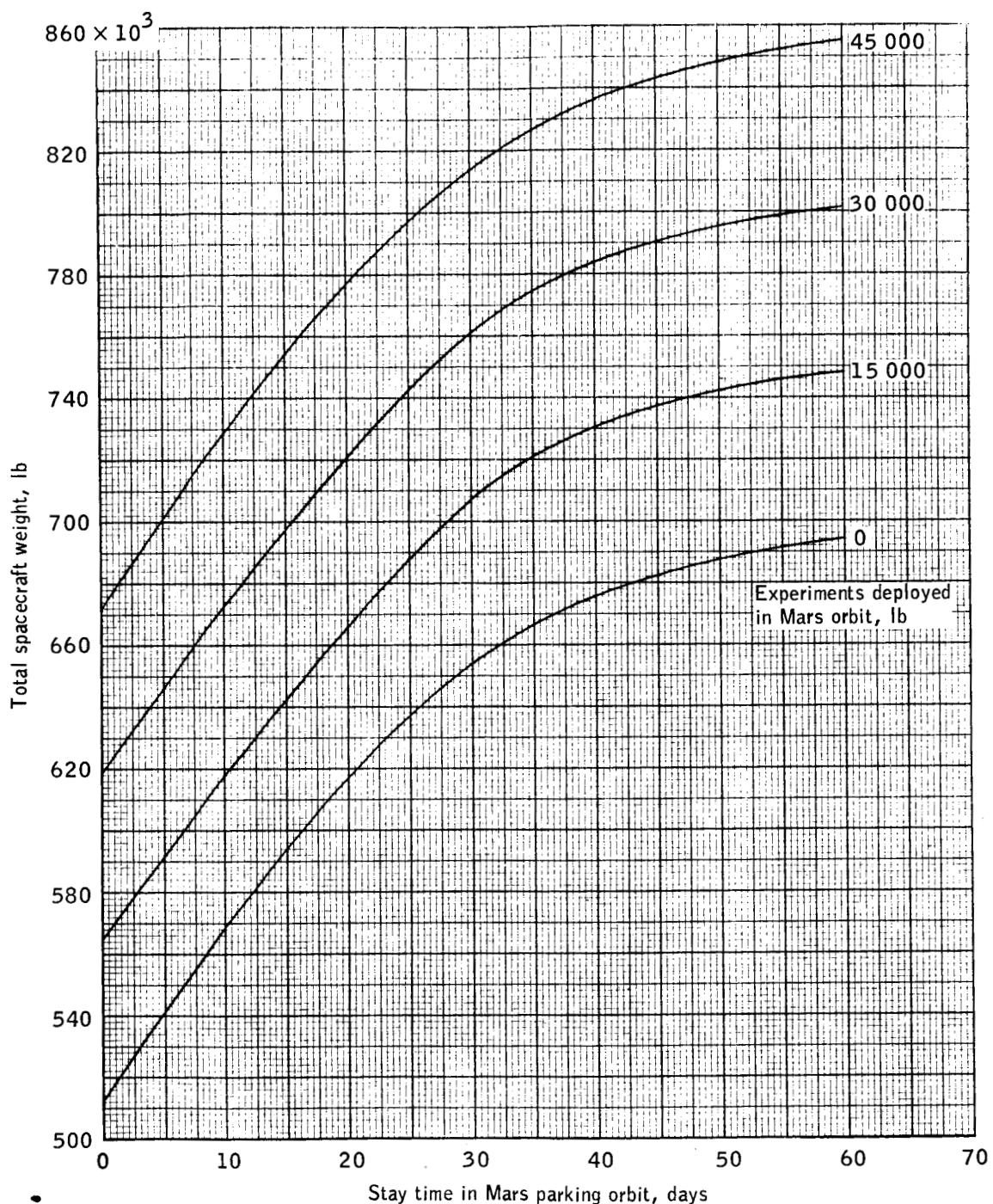


Figure 9.- 1983 Mars orbiting mission using Venus outbound swingby. Total mission duration is 600 to 640 days.

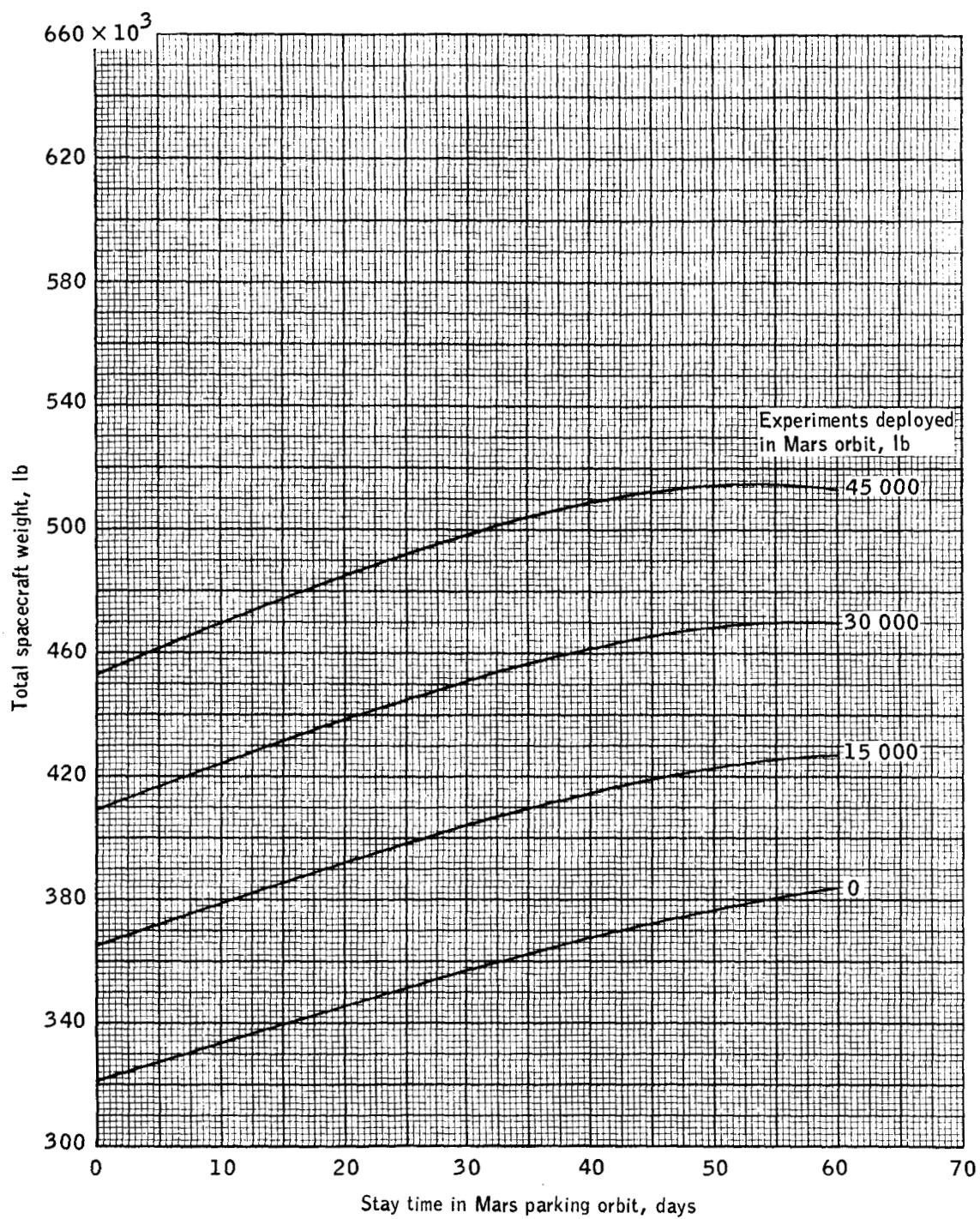


Figure 10.- 1985 Mars orbiting mission using Venus outbound swingby. Total mission duration is 560 to 680 days.

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